AERO31521
Conceptual Aerospace Systems Design

Individual System Design Report

Mission Analysis Engineer

Group E1

Student Name: Jiahao Geng

Student Number: 10683124

University of Manchester

Department of Mechanical, Aerospace and Civil Engineering

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Contents

1. Introduction

As a mission analysis engineer, the main responsibilities include Orbit selection, Launch & delivery, manoeuvres, and Propulsion system design. This individual report would demonstrate the design process and present developed concepts focusing on the above elements for the 'Mission' subsystem of an Earth observation satellite. Generally, the report would provide design options with the relative system model developments, system trade-offs, system sensitivity analysis, and feasibility analysis. In the end, the final solution choice would be made based on the analysis results to meet the top requirements with the relatively optimised performance.

2. Design Options

The top requirements for the Earth observation satellite are the ability to operate for 10 years and continuously monitor the specific region in China to track the sandstorm., and the frequency of the sensor capturing is about every15 minutes. The satellite is planned to be launched in 5 years and the Spacecraft Mass Budget is 3500kg (including a 15% system margin based on TRL). To achieve the top goal, the main requirements for the 'Mission' part are choosing a suitable orbit for the satellite operations, designing how to launch and delivery the satellites in a feasible way, designing the manoeuvres needed for the satellite to set in the chosen orbit, operate for 10 years and deorbit at the end of life, and designing the propulsion systems that could perform the manoeuvres.

2.1. System Model Development

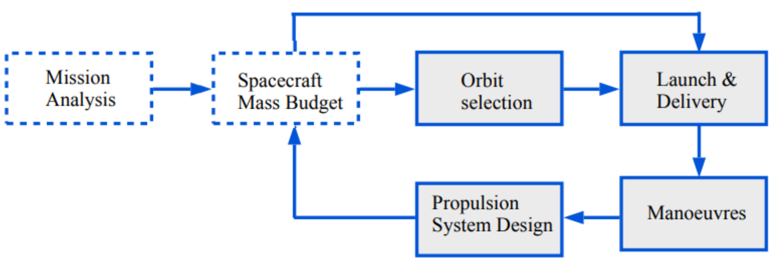

Figure 1. general system model for Mission analysis

2.1.1. Orbit selection

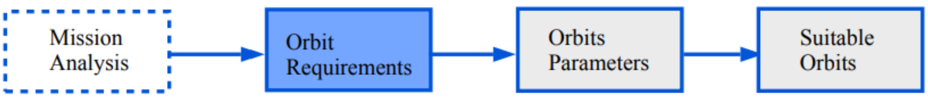

Figure 2. subsystem model for Orbit selection

The Mission Analysis based on the Top requirement of the satellite, which is the ability to monitor and track the sandstorm in the specific region in China continuously and capture the sensor image every 15 minutes. This gives the requirements for the orbit. The orbit should satisfy that 1 satellite on it could pass the region at least once in 15 minutes, to the other words, the period of the orbit should be 15 minutes. However, it is unachievable with only one satellite even in the low earth orbit (LEO), since the minimum period for LEO is still 90 minutes, and there is longitude difference due to self-rotation.

As a result, the key options considered are the geosynchronous orbit (GSO) and geostationary orbit (GEO). They both have the Period of 1436 minutes (one sidereal day), Eccentricity of 0, and Semi-major axis (SMA) of 42,164 km. The difference is that the GSO has non-zero inclination (Inc) whereas GEO is just above the equator (Inc: 0°). In principle, both GSO and GEO could satisfy the requirements, however, the ground track for the satellite on GSO would be an '8' shape, which means the passes is limited (instead of infinity for GEO), and the relative position between the satellite and the monitored region keeps changing, which might cause disturb for the data captured and extra energy consumption for attitude control, even though it could reduce the Inc change needed(Larson, 1999). So, GEO is initially accepted for the operation orbit.

The limitation in this part is that there no analysis about the detail influence of the position change caused by GSO's Inc.

2.1.2. Launch and Delivery

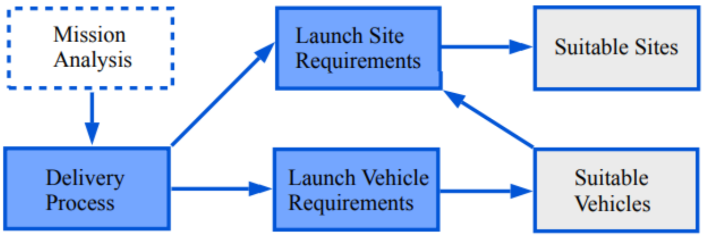

Figure 3. subsystem model for launch and delivery

Because the top goal is monitoring the sandstorms in China, the launch site and the launch vehicle are selected from the sites and rocket series manufactured from China, and the Mission Analysis is launching and delivering a satellite of around 3500kg from a launching site in China to GEO. For delivery process, Geostationary Transfer Orbit (GTO) is introduced. Using GTO could save the total energy needed from staging rocket (Larson, 1999). The general process is as following: At the beginning, the first and second sub-stages of the LM-3C/E rocket would send the combination of the satellite and the third sub-stage into a LEO (SMA:7000km) as circular parking orbit, then the third sub-stage performs a coasting flight of more than 600 seconds, and after the combination is reoriented under the control of the system, the third sub-stage engine fires again to send the combination into the target GTO orbit, and finally, the third sub-stage and the satellite are separated at the apogee of the GTO (r_p: 7000 km r_a: 42164 km). As a result, the selection of launch vehicle would base on the capacity of payload mass to GTO.

The key options for launch vehicle are the rocket LM-3C (3900kg GTO payload capacity) and LM-3B (5100kg GTO payload capacity). Since the initial satellite mass budget is 3500kg, the LM-3C is preferred at this stage because its GTO capacity is closer to the mass budget with some redundancy and prevent too much spare capacity as LM-3B (Krebs,2022).

The principles of choosing launch site are that: first of all, the launching the satellite must have the ability to launch the selected vehicle. Also, it is better to close to the equator, which could reduce the amount of inclination change required afterwards for the satellite and benefit more from the Earth's self-rotation. Moreover, the launch site should have water or deserts to the east to prevent any failed rockets from falling on a populated region. As a result, Wenchang Space Launch Site (Hainan, China) is chosen, since it can launch LM-3C and it is at low latitude (19° North), which could save energy for launching. It is also within 10 kilometres of the Pacific Ocean in the direction of launch. (Xue, 2021)

The Assumptions and limitations in this part is that it only considers the situation when the launch vehicle just takes the orbit change to GTO, and the satellite itself would do the inclination change and from GTO's apogee to GEO. But another potential solution is that the third stage could also perform a part of inclination change for the satellite (Thomas, 2016). For example, if the third stage changed 10° of Inc at the first burn, then the satellite only needs change the rest 10° of Inc instead of 20°, which could the reduce the corresponding ΔV and propellant needed for the satellite. However, since this situation relates to the detailed analysis of the rocket, it would not include in this report.

Another key option is increasing the perigee of the GTO, this could slow down the velocity at the apogee and since reduce the ΔV needed for satellite to GEO. However, it depends on the ability of the first & second stage of the launch vehicle, and this is also difficult to analyse in this report.

2.1.3. Manoeuvres

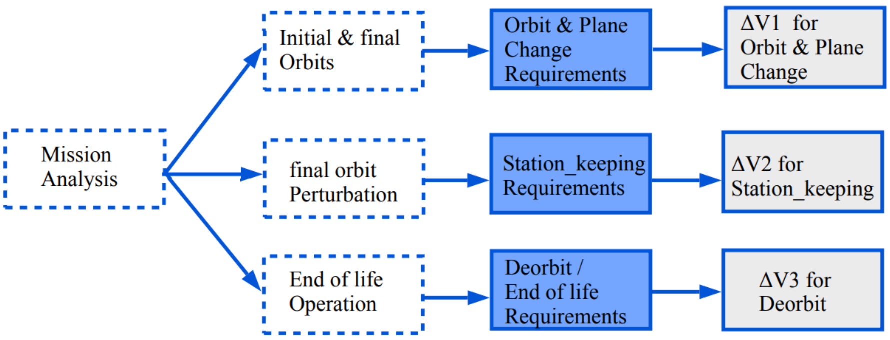

Figure 4. subsystem Model for Manoeuvres

The mission analysis for Manoeuvres is generally about the ΔV values for 3 parts: Orbit change from GTO to GEO (including a plane change), Station-keeping in GEO for 10 years, and Orbit change from GEO to the Graveyard Orbit (SMA around 42500km).

For GTO to GEO: there are 2 key options: combine the orbit change and Inc change together and another is performing them separately.

For combined operation: $\Delta V_1 = \sqrt{v_{T,a}^2 + v_{GEO}^2 - 2v_{T,a}v_{GEO}\cos(\Delta i)} = 1.633 \text{ km/s}$

For separate operations: $\Delta V_1 = v_{GEO} - v_{T,a} + 2v_{GEO}sin(\Delta i/2)$ = 2.001 km/s

Where, $v_{\mathrm{T},a} = \sqrt{\frac{2\mu_{Earth}}{r_a} - \frac{\mu_{Earth}}{a_{\mathrm{t}}}}$, $v_{\mathrm{GEO}} = \sqrt{\frac{\mu_{Earth}}{r_{\mathrm{GEO}}}}$

$v_{T,a}$ is the velocity at the Apogee of the GTO.

μ_{Earth} is the gravitational parameter of Earth [km^3/s^2], the value is 398600.

r_a is the apogee of the GTO, here is 42164 km. a_{t} is the Semi-major Axis of the GTO, here is 24582 km

v_{GEO} is the velocity at the GEO .r_{GEO} is the radius of the GEO, here is 42164 km.

Δi is the inclination difference between the GTO and GEO, here is 20°.

Under this setting, the combined operation is selected because of its lower ΔV .

For Station-Keeping, the ΔV mainly comes from the manoeuvre needed to counter the precession motion caused by a combination of lunar gravity, solar gravity, and the flattening of the Earth at its poles, which is an initial inclination gradient is about 0.85° per year and amounting to a ΔV of approximately 0.05 km/s per year (Anderson, 2015). A second effect to be taken into account is the longitudinal drift, caused by the asymmetry of the Earth. Any geostationary object placed between the equilibrium points (at 75.3°E and 108°W) would be slowly accelerated towards the stable equilibrium position, causing a periodic longitude variation (Friesen,1992). The correction of this effect a maximal ΔV of about 0.002 km/s per year, depending on the desired longitude (Anderson, 2015). Solar wind and radiation pressure also exert very small forces on the satellite, which would cause slowly drift away from their prescribed orbits over time (Kelly, 2016). Overall, since the designed working life for the satellite is 10 years, the total ΔV_2 for station-keeping is about 0.53 km/s

For GEO to the Graveyard Orbit: $\Delta V_3 = \sqrt{\frac{\mu_{Earth}}{r_{GEO}}} - \sqrt{\frac{\mu_{Earth}}{r_{\mathrm{GyO}}}}$ = 0.011 km/s.

Where, r_{GyO} is the radius set for the Graveyard Orbit, here is 42464 km, based on the U.S. government regulation which states that the altitude of graveyard orbit should be around 300 km away from the GEO belt. It also includes the margin for the influence of the solar radiation pressure and other effects. (Kelly, 2016)

Apply relative margins based on 'Margin philosophy for science assessment studies', ESA/ESTEC:

ΔV_1 would be 1.71 km/s (5%), ΔV_2would be 1.06 km/s (100%), ΔV_3 would be 0.012 (5%)

For this part, the limitation and assumption are mainly for the direct use of external data instead of detail analysis based on the satellite itself like mass and aspect area etc.

2.1.4. Propulsion System design

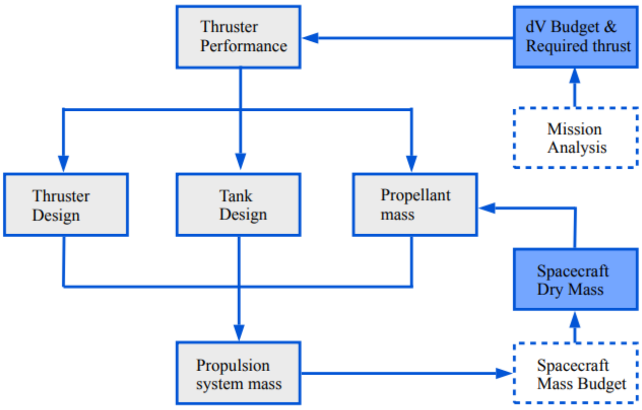

Figure 5. subsystem Model for propulsion

The mission analysis for this part is that there would be 2 propulsion system used on the satellite to complete the manoeuvres mentioned above, for the 10-year lifetime at 42164 km GEO. The initial dry mass is 2000kg.

One is a chemical propulsion system, using MMH and MON propellant thruster with mass of 4.5kg, force of 400 N and a specific impulse (Isp)of 450 s (referred on s400 motor from Ariane Group), basically for the orbit transfer from GEO apogee to GEO for the satellite.

Another is an electrical propulsion system, using Xeon propellant ion thrusters with mass of 14kg, force of 0.11 N and an Isp of 3400 s, (referred on XIPS-25 from L-3 Communications), basically for the station-keeping and deorbit to the Graveyard orbit at the end of life.

For mass of propellant, Using the ideal rocket equation: $\Delta V = I_{sp} g_0 \log \frac{m_i + m_p}{m_i}$

For mass of tanks, using formulae: $V_{tank} = \frac{m_p * R * T}{P_{max} M_P}$ $r_{tank} = \sqrt[3]{\frac{V_{tank}}{\frac{4}{3}\pi}}$

$t_{wall} = FoS * \frac{P_{max} r_{tank}}{2\sigma_{yield}}$ $m_{tank} = \frac{4}{3}\pi[(r_{tank} + t_{wall})^3 - r_{tank}^3]\rho_{tank}$

Where, P_{max} is 7 Mpa, m_p is propellant mass, M_P is molar mass of propellant,

σ_{yield} is 1230Mpa, ρ_{tank} is 4460 kg/m^3, Factor of Safety (FoS) is 2.

Assume the tank is spherical and used Titanium as material. Also, apply a margin of 2% for propellant residual (ESA/ESTEC, 2012)

After several iterations, the mass for chemical propulsion system would be 1492.2 kg

(935 kg for propellant, 493 kg for tank, 4.5 kg for thruster)

the mass for electrical propulsion system would be 91.24 kg

(61.7 kg for propellant, 15.54 kg for tank, 14 kg for thruster)

The total mass of the propulsion system is 1583.44 kg

Another key option for the propulsion system design is that only use the electrical propulsion for all the manoeuvres. Because the Isp of ion thruster is much higher than the chemical thruster, the required total mass of propulsion system would be much lower than the result above (Thomas, 2016). The propulsion system mass for this option is only about 235 kg. However, the force generated by the ion thruster is much lower than the chemical thruster, to the other words, the total transfer time for the satellite is much longer, which is over 300 days compared to several hours using chemical thruster (Thomas, 2016). To let the satellite begin operation as soon as possible, the chemical thruster is kept.

A limitation for the analysis in this part is that the chemical propellants is considered as a mix, which might cause an error on estimation of molar mass and then the propellant mass.

2.2. System Trade-offs

2.2.1. orbit selection

The GSO with relatively small inclinations still 3 could be accept instead of the GEO. There is a trade-off between the inclination change needed and the precision & quality of capturing. For every degree of Δi reduced for the satellite, there is a reduction on ΔV for around 17m/s.

2.2.2. launch and delivery

A trade-off could be made between the increasing of GTO perigee and the energy consumption by the first & second stages of rocket (within the limit). For every 500 km increased at perigee, there is a reduction on ΔV for around 35m/s. Another trade-off could be made between the Δi changed by the third stage and its own energy consumption (or payload capability). For every degree of Δi reduced for the satellite, there is a reduction on ΔV for around 17m/s.

The TRL for the selected launch site and vehicle are both at 9, which means actual system proven in operational environment and the relative data is trustable. For future development. There might be a sea-platform launch site at equator available, or a vehicle developed with ability to get higher perigee with same payload capability. They could effectively reduce the Δi needed for the satellite.

2.2.3. manoeuvres

This part is basically at TRL 2. The data from literature also mainly from theoretical simulations. Future development would be using the data collected directly from the real satellites in the similar circumstances.

2.2.4. propulsion system design

There could be a trade off between the Factor of Safety and the propellant tank mass. Even though increase FoS could enhance the Reliability, the tank mass would significantly rise. For example, if increase FoS from 2 to 3, the tank mass for the chemical propellant would rise from 462.7 kg to 695 kg.

For this part, the TRL of the propulsion systems are at 2, because the data used are driven from existing thrusters, which represents an estimation of future development on the thruster performances. The value of Isp would have big influence on the propellant mass estimation, especially for the chemical thruster.

2.3. System Sensitivity Analysis

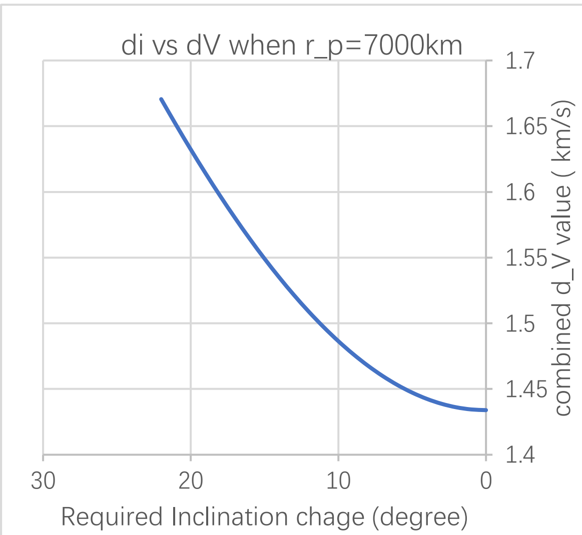

Figure 6. Δi vs ΔV when r_p of GEO = 7000km

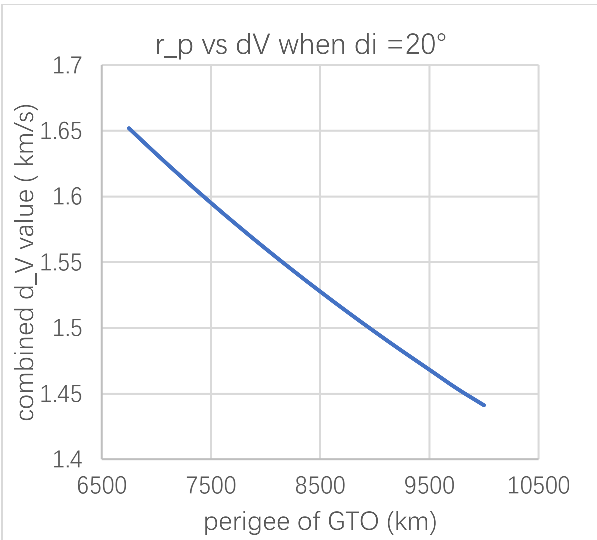

Figure 7. r_p vs ΔV when Δi = 20°

As the figures 6,7&8 shown, decreasing the required inclination change or increasing the perigee are both able to reduce the ΔV requirement for the satellite. And the combination of the alternatives as mentioned in previous parts would have a more effective result in the reduction of ΔV for orbit and plane change, to the other words, the requirement of propellant mass on satellite decreased.

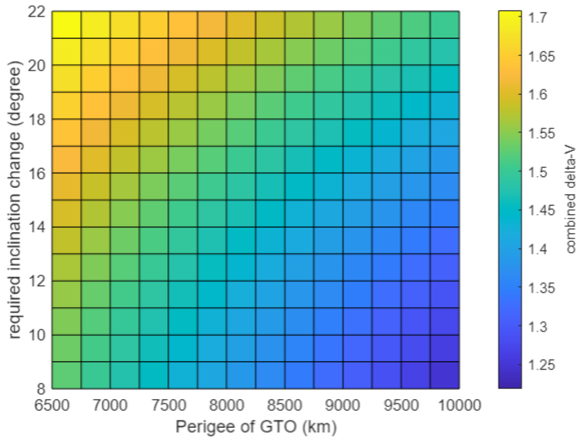

Figure 8. The effect for combination of changing Δi and r_p simultaneously

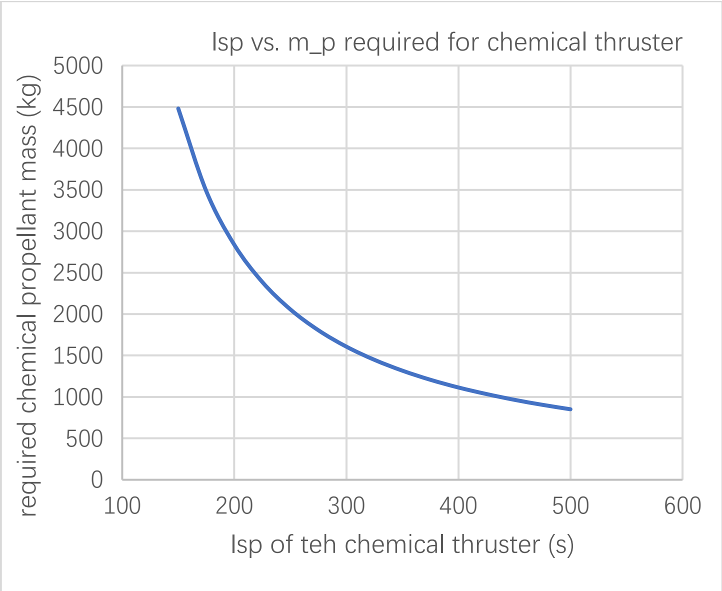

Figure 9. The relation between chemical thruster Isp and required propellant mass for ΔV =1710 m/s

As the figure shown, the Isp of chemical thruster is very important to the propulsion system, especially for Isp that is lower than 300s. At this stage, a small increase in the Isp for thruster would lead to a huge reduction in the chemical propellant mas required for a given ΔV.

2.4. Feasibility Analysis

For orbit selection, by applying the system model, the orbit requirements could be decided from the top goal. Then, the suitable orbits would be found out by matching the orbit parameters with the requirements. Firstly, the GEO is selected since it perfectly satisfies the requirements. Then the GSO with small Inc is also chose as alternative since it has the same period, and the influence of Inc is acceptable. As a result, if the satellite could not complete a plane change to the GEO, it is still able to operate in GSO.

For launch and delivery, the feasibility is guaranteed since the selection of launch site, vehicle and delivery process through GTO are all in RTL-9. And a sufficient of spare capacity is applied to counter the increase in payload.

For Manoeuvres, the model is established to calculate the ΔV at different stages. Since the data are get from technical literature and relative margins are applied based on the instruction of ESA/ESTEC, the results should be reliable.

For propulsion systems, the model is built including an iteration loop to estimate the system mass required to satisfy the satellite manoeuvres. By applying the same formulae to the alternative of only using electrical thruster and reviews from other literature, the feasibility is established. The pros and cons of chemical and electrical thruster are also found and compared from the model.

2.4.1. Environmental and Social Impact

To prevent the hazard that might caused by a failed launch of rocket, the launch sites are usually built away from the area with crowded populations. And one reason of choosing Wenchang Space Launch Site is because it is close to the ocean. However, if the failed rocket falls into the sea, the rest propellant might cause serious pollution since it is usually toxic.

Based on the regulation from USA government and Inter-Agency Space Debris Coordination Committee (IADC), the end-of-life operation is designed to transfer the retired satellite from GEO to the graveyard orbit with SMA of 42464 km. since it is far away from the GEO, it would be unable to cause disruption to other operating GEO satellites. Since the two orbits are far away from earth, and the manoeuvre is done by electrical thruster, the cost of propellant is very small.

If choosing the option of only using the electrical propulsion, the low thrust finite burn would make the satellite pass through the area of LEO and MEO lots of time, which might cause disturbances to other satellites operating in these orbits. That is why it is just an alternative.

2.4.2. Risk Analysis

The risk during launch and delivery is the possibility of fail due to some factors like disfunction of some key components, or wrong operations caused by human errors. The weather sometimes is also an important factor for launch success, since the wind near the sea is usually very big.

For the alternatives of using longer perigee of GTO and performing part of inclination change by the rocket stage, there is a risk of lack of energy. It might cause the satellite to use more energy to correct the orbit and a reduction in lifetime. So a lower perigee with no plane change would be more reliable.

For Manoeuvres, a possible risk is failed to keep the correct attitude and cannot get into the right orbit. Another is the disfunction of thruster or leak of propellant, they would cause the lack of energy for the satellite to perform the right manoeuvre. The solar storm in space is also a risk during the station-keep period, which might cause damage of component.

3. Conclusions

Based on the system model established and analysis on selected options and alternatives, the summary of the mission analysis would be as following:

Operation orbit: GEO, satellite stays at 100°E.

Launch site: Wenchang Space Launch Site (Hainan, China)

Launch vehicle: Long March 3 C/E (CZ-3C/E)

Delivery: Through Low Earth Orbit (LEO, SMA: 7000 km, Inc: 20°) and Geostationary Transfer Orbit (GTO, perigee: 7000 km, apogee: 42164 km, SMA: 24582 km, Inc: 20°) with staging rocket.

Propulsion systems on satellite: Chemical Propulsion system (1492.2 kg) and Electrical Propulsion system (91.24 kg)

End-of-life operation: move to the Graveyard orbit (SMA of 42464 km)

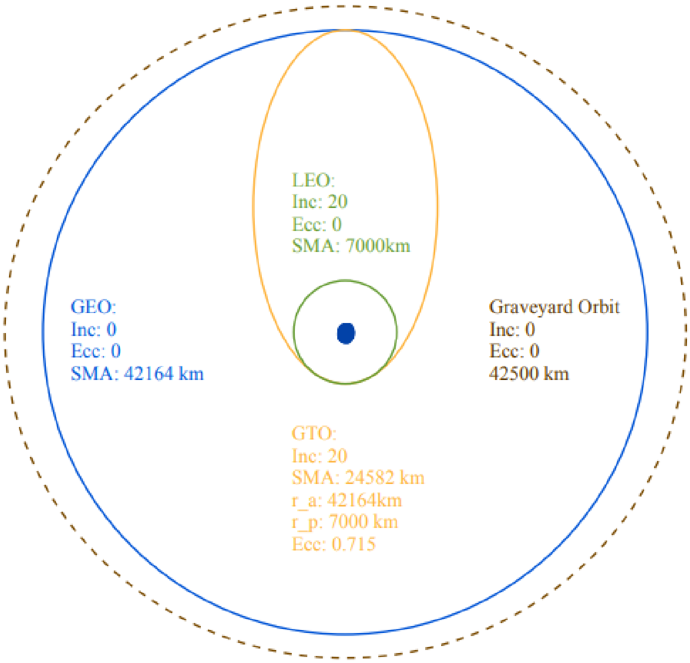

Figure 10. sketch of orbits included in the mission

4. References

Anderson, P.V. *et al.* (2015) ‘Operational considerations of GEO debris synchronization dynamics’, in *66th International Astronautical Congress, IAC-15 A*. All Rights Reserved, p. 7.

Byers, D.C. and Dankanich, J.W. (2008) ‘Geosynchronous-earth-orbit communication satellite deliveries with integrated electric propulsion’, *Journal of Propulsion and Power*, 24(6), pp. 1369–1375.

Chiasson, T.M. (2012) *Modeling the characteristics of propulsion systems providing less than 10 N thrust*. Massachusetts Institute of Technology.

Friesen, L.J. *et al.* (1992) ‘Analysis of orbital perturbations acting on objects in orbits near geosynchronous Earth orbit’, *Journal of Geophysical Research: Planets*, 97(E3), pp. 3845–3863.

Han, D.I., Han, C.Y. and Shin, H.D. (2009) ‘Empirical and computational performance prediction for monopropellant hydrazine thruster employed for satellite’, *Journal of Spacecraft and Rockets*, 46(6), pp. 1186–1195.

Hudson, J.S. and Kolosa, D. (2020) ‘Versatile on-orbit servicing mission design in geosynchronous earth orbit’, *Journal of Spacecraft and Rockets*, 57(4), pp. 844–850.

Kelly, P. *et al.* (2016) ‘Solar radiation pressure applications on geostationary satellites’, in *Proceedings of the 2016 AAS GP & C Conference. American Astronautical Society*.

Krebs, G.D. (2022) ‘“CZ-3C (Chang Zheng-3C)”. Gunter’s Space Page.’ Available at: https://space.skyrocket.de/doc_lau_det/cz-3c.htm.

Larson, W.J. and Wertz, J.R. (1999) ‘Space mission analysis and design’, *Space*, 5(14), p. 110.

SRE-PA & D-TEC staff (2012) ‘Margin philosophy for science assessment studies’.

Thomas, D. (2016) *A Comparison of GEO Satellites Using Chemical and Electric Propulsion*.

Xue, Y. *et al.* (2021) ‘Rocket launching activities are associated with reduced insect species richness and abundance in two types of tropical plantations around the Wenchang Satellite Launch Center, southern China’, *Ecological Indicators*, 127, p. 107751.